Final Report: September 12, 1996

National Aeronautics and Space Administration (NASA) DC-XA Clipper Graham Mishap Investigation Report

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*Transmitted under separate cover to the Director, Safety and Risk Assessment Division, Office of Safety and Mission Assurances, NASA Headquarters -- Do not duplicate or distribute.

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SECTION 1

TRANSMITTAL LETTER

To: NASA Headquarters
Attn: Q/Associate Administrator for Safety and Mission Assurance

From: Chairperson, DC-XA Mishap Investigation Board

Subject: DC-XA Mishap Investigation Report

In accordance with your memorandum of August 12, 1996, the subject report has been completed and signed by the board members. Fifteen copies of the report are hereby forwarded to you in accordance with NASA Management Instruction 8621.1F, Mishap Reporting and Investigation.

Vance D. Brand

Enclosure

SECTION 2

SIGNATURE PAGE

NASA DC-XA MISHAP INVESTIGATION BOARD

____
Vance D. Brand, Chairperson
Dryden Flight Research Center

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Charles E. Harris
Langley Research Center

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George D. Hopson
Marshall Space Flight Center

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David C. Sharp, Lt Col, USAF
Air Force Safety Center, Kirtland AFB NM

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Warren I. Wiley
Kennedy Space Center

SECTION 3

LIST OF MEMBERS, ADVISORS, OBSERVERS, AND OTHERS

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Members:

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George D. Hopson, Deputy Director, Science and Engineering, Marshall Space Flight Center, AL

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Executive Secretaries:

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Kay George, White Sands Missile Range, NM

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Judy Duffield, Dryden Flight Research Center, CA

SECTION 4

EXECUTIVE SUMMARY

On July 31, 1996, at 13:15 MDT, NASA and McDonnell Douglas successfully launched and flew the Clipper Graham, DC-XA, vehicle for the fourth time. Following an uneventful takeoff, the Clipper Graham flew for 104 seconds reaching an altitude of 4100 feet and traveling 2800 feet up range before returning to the concrete landing pad, successfully completing all test objectives.

Ninety-eight seconds into the flight and at approximately 400 feet, the DC-XA computer commanded landing gear deployment. Over the next five seconds, three of the four legs successfully deployed. Four seconds after the gear deploy command, landing gears 1 and 4 deployed within one-tenth of a second of each other. Then, landing gear 3 deployed one full second later. Landing gear 2 never deployed. Descending from 400 feet, the spacecraft landed safely on three of its four legs. When the weight-on-gear indication was received at 13:17:27 MDT, the engines terminated as programmed and at 13:17:29 MDT, the vehicle toppled toward the position of landing gear 2.

Upon impact, the vehicle was destroyed in a series of three explosions spaced over the next 90 seconds. The first explosion at 13:17:30 MDT ignited the composite shell and the avionics rack. At 13:17:40 MDT, ten seconds after the initial explosion, the fire suppression system began dispensing water. A second explosion of liquid oxygen from the aluminum-lithium tank rocked the mishap scene ten seconds after the first explosion. The fire suppression system shut down after the tank ran out of water but before complete fire extinction. Approximately 1 minute after the second explosion, the hydrogen tank exploded. This third explosion scattered the composite material from the aeroshell and hydrogen tank over the mishap scene.

The Clipper Graham DC-XA vehicle was totally destroyed by ground impact and ensuing explosions and fires. The vehicle struck the ground on the corner of the vehicle at the undeployed landing gear 2. The upper two-thirds of the composite aeroshell, aluminum-lithium liquid oxygen tank, composite liquid hydrogen tank, composite intertank, avionics, nose cone, and parachute recovery systems were destroyed during the three explosions and ensuing fire. Parts of landing gears 3 and 4 were melted in the fire as well. Landing gears 2 and 1 mechanisms were damaged during the tip over and ensuing fires and explosions. The lower one-third of the vehicle aeroshell, containing the throttlable RL-10 engines and the auxiliary propulsion system, were charred and covered with soot. The RL-10 engines and auxiliary propulsion system were the only items appearing to be recoverable.

Videotapes of the flight and still photographs of the wreckage showed that landing gear 2 failed to deploy. This failure was also evident in the helium supply pressure time history. NASA analysts at Kennedy Space Center (KSC) performed a helium pressure decay study and showed the loss of helium to be greater than expected after the start of gear deployment. Also, they estimated the diameter of the hole in the pneumatic system that would be required to achieve the observed helium pressure decay rate. This analysis indicated that if the brake line was disconnected during landing gear deployment, the decay rate would be equivalent to that which was recorded by the flight instrumentation. Also, post mishap inspection found the landing gear to be stowed and the pneumatic brake line not connected.

Therefore, the primary cause of the vehicle mishap was that the brake line on the helium pneumatic system for landing gear 2 was not connected. This unconnected brake line prevented the brake mechanism from being pressurized to release the brake and resulted in landing gear 2 not extending. The vehicle became unstable upon landing, toppled onto its side, exploded, and burned.

Contributing causes of the mishap were as follows:

- Design of the system for gear stowage required technicians to break the integrity of the helium brake line after integrity had been already verified. No other check was conducted to reverify the integrity of the system after disconnection and reconnection of the line was completed.
- Landing gear stowage was never identified as a critical process. No special steps were taken to ensure the readiness of this system for flight.
- During the gear stowage process, there was no record of checking off steps or evidence of cross-checking of work by another person.
- Distraction or interruption of the mechanical technician during gear stowage operations may have contributed to the nonconnection of the brake line.

The design of the DC-XA vehicle and operational procedures were driven by rapid development and low cost. Accordingly, a minimum number of personnel were involved in operations. Also, design was single string, and there was just one flight test vehicle. There was strong reliance on good people but not a lot of margin for human error afforded by the vehicle preparation process. The McDonnell Douglas Rapid Prototyping Guidelines or implementation thereof for the DC-XA may have gone too far in the direction of sacrificing quality and reliability. This rapid prototyping concept should be revisited from an operations perspective.

SECTION 5

METHOD OF INVESTIGATION, BOARD ORGANIZATION, AND/OR SPECIAL CIRCUMSTANCES

5.1. BACKGROUND

The DC-XA mishap occurred at approximately 13:17 MDT, July 31, 1996. (See NASA Mishap Report in appendix A).

Immediately after the mishap, White Sands Missile Range, Phillips Laboratory, McDonnell Douglas Aerospace, and NASA began gathering perishable data which included telemetry, radio communications, optical tracking, vehicle documentation, ground crew and witness statements, and crash site video and photo documentation. In addition, the project team took necessary actions to safeguard the mishap site and obtain other information which would be required by the mishap investigation and which could be obtained without disturbing or destroying evidence.

The process of establishing the formal mishap investigation board was initiated on August 1, 1996, by the Associate Administrator for Safety and Mission Assurance at NASA Headquarters in response to a request from the Associate Administrator for Space Access and Technology. The memorandum establishing the board (appendix B) was signed on August 12, 1996. The investigation board commenced its activities at White Sands Missile Range, New Mexico on August 5, 1996.

5.2 METHOD OF INVESTIGATION

The DC-XA Mishap Investigation Board convened on August 5, 1996, at White Sands Missile Range, New Mexico. A Chairman and four Board Members were appointed from various disciplines, NASA Centers, and the U.S. Air Force. Several advisors and consultants were also appointed to assist in the investigation. Initial task assignments included visiting the mishap site, interviewing witnesses, obtaining video tapes and single frame photograph hard copies, requesting additional photography and videography, monitoring removal of the suspect landing gear 2 pneumatic panel, and obtaining and evaluating flight and landing data. Subsequently, the Board reviewed and evaluated the DC-XA hardware checkout and operating procedures.

The DC-XA contractor’s project manager and technical personnel provided an overview of the DC-X and DC-XA program, specific aspects of the DC-XA landing gear system, and preliminary insight into a potential cause. The data, video tapes, and single frame photographs had been impounded by the McDonnell Douglas Aerospace DC-XA Project for use by the Board.

The vehicle was left in place after the mishap and fire containment. The area had been foamed and later sprayed with a wax solution to seal potentially harmful composite particulates. The vehicle had been made safe by a McDonnell Douglas and U.S. Army team that also conducted a very brief and preliminary assessment of the hardware conditions. After the Board conducted a detailed review of the launch vehicle mishap site, it considered the process it would use to determine the primary cause of the mishap, contributing causes, and recommendations. The Board considered attempting to operate landing gear 2 in-place at the mishap site; however, after discussions and a meeting with McDonnell Douglas personnel, this approach was not considered feasible.

During subsequent days, all sources of data were examined and possible failure causes were discussed. The Board compiled a sequence of events from witness statements, flight data, and flight videos. In addition, the landing gear 2 pneumatic panel was removed from the wreckage and visually examined by the Board.

At the Board's request, McDonnell Douglas provided a simplified landing gear deployment fault tree of sufficient depth to highlight potential failures that could have led to the mishap. Landing gear 2 was removed from the vehicle and shipped to McDonnell Douglas-West Aerospace in Huntington Beach, California, for functional testing and evaluation. A board member witnessed this testing.

Findings and recommendations were then prepared by the Board and can be found in Section 8 of this report.

SECTION 6

NARRATIVE DESCRIPTION

6.1. NARRATIVE DESCRIPTION OF MISHAP

During the afternoon of July 29, 1996, mechanical technicians tested the deployment of DC-XA landing gear using DCXP-002 procedure (ref. 1), page 3-08A-19, steps 3-08A-090 through 3-08A-95, and installed the avionics camera. The test and installation procedures went successfully.

The next morning, July 30, 1996, the DC-XA team was locked out of the White Sands Missile Range (WSMR) until 09:00 MDT for range activities and a range coordination meeting. The procedures in the pad area began at approximately 10:00 MDT with the charging of the helium and nitrogen ground supply tanks from low pressure supplies. Between 10:30 and 12:30 MDT, the propellent management team started loading the liquid hydrogen and liquid oxygen storage tanks. At 13:30 MDT, maintenance technicians and supervisors attended a preflight briefing by the flight manager in the Operations Trailer. At 14:00 MDT, the landing gear retraction (DCXP-002 Procedure, page 03-08A-19, steps 3-08A-96 through 3-08A-110) was started with landing gear 3 being successfully retracted. The positioning of personnel for this procedure was as follows: maintenance technician (MT)1 was on the Marklift work platform approximately 15 feet off the ground. MT2 entered the boattail through the access panel behind the flap for access to the landing gear's unlocking cylinder. Meanwhile, MT3 and MT4 were below the launch vehicle helping to position the foot pad onto the pins on the bottom of the heat shield and to remove and install the landing gear's unlocking cylinder set screws. All four maintenance technicians then moved to landing gear 2. During stowage of landing gear 2, MT1 may have been distracted or interrupted. In any case, MT1 failed to complete step 3-08A-108 of the procedure. Before installation of the access panel and completion of landing gear 2 stowage, step 3-08A-110, the Marklift work platform was diverted to complete troubleshooting of the camera in the avionics rack on the same side of the vehicle by another maintenance technician. Once that camera troubleshooting was complete the Marklift work platform was moved to the other side of the vehicle for stowing of landing gears 1 and 4. Concurrent with the just described procedure, a team of personnel from the Marshall Space Flight Center began the Optical Plume Anomaly Detection system calibration that required clearing the vehicle of personnel, activating the flaps, and running the hydraulics to center the engine gimbals.

On July 31, 1996, at 13:15 MDT, NASA and McDonnell Douglas successfully launched and flew the Clipper Graham DC-XA vehicle. Following an uneventful takeoff the Clipper Graham flew for 104 seconds reaching an altitude of 4100 feet and traveling 2800 feet up range before returning to the concrete landing pad and successfully completing all test objectives (Appendix C).

During descent at 98 seconds into the flight, the command was issued for landing gear deployment. Over the next 5 seconds three of the four legs successfully deployed. Four seconds after the signal was sent, landing gears 1 and 4 deployed within one-tenth of a second while landing gear 3 deployed 1 full second later. Landing gear 2 never deployed. Descending from 400 feet, the spacecraft landed on its three deployed legs. When the weight-on-gear indication was received after a 2 1/2 minute flight, the engines terminated as programmed, and the vehicle, now unstable, toppled toward the landing gear 2 position. (Photographs in appendix G show this sequence of events.)

Upon impact, the vehicle was destroyed in a series of three explosions spaced over the next minute and a half. The first explosion, probably caused by rupture of the liquid oxygen tank, ignited the composite shell and the avionics rack. Ten seconds after the initial explosion, the fire suppression system began dispensing water. This system shut down after the tank ran out of water but before complete fire extinction. The second explosion of the liquid oxygen from the aluminum-lithium tank rocked the mishap scene 20 seconds after the first explosion. Approximately 1 minute after the second explosion, the third and final explosion rocked the accident scene when the hydrogen tank exploded scattering the composite material from the aeroshell and tank over the mishap scene.

During the fire fighting, six optical video camera technicians drove through the smoke, inhaling the composite dust and smoke from the fire. They were treated by the ambulance technicians on the scene and released. No other personnel were exposed to the smoke or composite materials.

The White Sands Missile Range Fire Department applied fire suppressant foam on the wreckage at 15:00 MDT. The WSMR Fire Chief declared the fire extinguished at 17:00 MDT. The spacecraft composite shell, aluminum-lithium oxygen tank, composite hydrogen tank, and avionics rack were destroyed. Portions of the RL-10 engines and auxiliary propulsion system survived and are potentially salvageable but will require major refurbishment.

The site was made safe by explosive ordinance disposal and McDonnell Douglas personnel wearing hazardous material suits on August 1, 1996. An 18-inch high gypsum earth berm was erected around the mishap scene to control any further contamination from the composite materials. At 16:00 MDT, hazardous material personnel started the post mishap cleanup of the site beginning with the spraying of the composite wreckage with a water and wax mixture creating a barrier to contain hazardous carbon fiber dust.

On August 2, 1996, McDonnell Douglas personnel were allowed to enter the site to insert desiccants into the engines, and they finished taping the chilldown ducts to protect the engines from the weather and blowing gypsum.

The chronology of the mishap sequence and resulting destruction of Clipper Graham is as follows:

TIME (MDT)	TIME (UTC)	EVENT
13:15:00	19:15:00	Launch actions start
13:15:03	19:15:03	Ignition and liftoff (nominal)
13:17:11	19:17:11	Deploy landing gear at approximately 400 feet
13:17:11	19:17:11	Deployment of landing gears 1, 4, then 3 at approximately 300 feet
13:17:16	19:17:16	Flight Operations Control Center noted that a landing gear failed to deploy
13:17:24	19:17:24	Successful touchdown
13:17:27	19:17:27	Weight on gear and engine termination

13:17:29	19:17:29	Vehicle topples toward landing gear 2
13:17:30	19:17:30	Explosion 1
13:17:40	19:17:40	Ground-based fire suppression system engaged until out of water
13:17:50	19:17:50	Explosion 2
13:18:54	19:18:54	Explosion 3
17:00	23:00	Fire Chief declares fire extinguished

6.2 FACTUAL DESCRIPTION OF VEHICLE DAMAGE

6.2.1. GENERAL

The Clipper Graham, DC-XA, vehicle was totally destroyed by ground impact and the ensuing explosions and fires. The vehicle struck the ground near the corner where undeployed landing gear 2 was located. The upper two-thirds of the composite aeroshell, aluminum-lithium liquid oxygen tank, composite liquid hydrogen tank, composite intertank, avionics, nose cone, and parachute recovery system were destroyed during the three explosions and ensuing fire. Parts of landing gears 3 and 4 were melted in the fire as well. Landing gears 2 and 1 mechanisms were damaged during the tipover and ensuing conflagration. The four RL-10 rocket engines and the auxiliary propulsion system (APS) were only slightly coated with soot, and they appeared to be recoverable. (Photographs in Appendix G show various views of the wreckage.)

6.2.2. AEROSHELL AND NOSE CONE

The aft and forward graphite/epoxy and foam core sandwich aeroshells were split approximately halfway down the body and completely charred by the fire and explosions. The top two-thirds of the aeroshell were melted and shattered beyond repair. The graphite/epoxy and fiberglass nose cone was split lengthwise three-fourths the length from the base to the tip.

6.2.3. AVIONICS RACK

The avionics rack equipment, batteries, and two helium storage bottles were destroyed by the fire and explosions. None of this equipment is salvageable.

6.2.4. OXYGEN TANK

The Russian-built aluminum-lithium liquid oxygen tank partially split in the impact with the ground and with the second explosion. The upper welds split and the internal slosh baffles deformed with the resulting fire and third explosion.

6.2.5. INTERTANK

The DC-XA intertank is constructed of graphite/epoxy material. The intertank was destroyed in the explosions and resulting fires and was not salvageable.

6.2.6. HYDROGEN TANK

The Clipper Graham graphite/epoxy shell of the hydrogen tank split during the explosions and fires. The tank was split in two near the top of the tank. The hydrogen explosion separated the top half of the wreckage approximately 6 feet from the bottom half of the vehicle. Fire burned the outside of the tank and destroyed some of the tank's internal insulation tiles.

6.2.7. THROTTLABLE RL-10 ENGINES

Four throttlable RL-10 engines using liquid hydrogen fuel and liquid oxygen propelled the vehicle. These engines gimballed to control rotation and translation. Although singed and soot covered, the engines appeared to be recoverable.

6.2.8. AUXILIARY PROPULSION SYSTEM

The Russian-built auxiliary power unit is the heart of the APS. Composite liquid hydrogen lines and valves connect the liquid hydrogen tanks to a gas converter system. The integrated APS is designed for gaseous hydrogen propellant generation for the reaction control system and to provide another power source for hydraulics which enable engine gimbal and flap actuation during flight testing. The system was disconnected and drained for this flight. The APS may be salvageable.

6.2.9. LANDING GEAR

The German-designed and built landing gear consists of pneumatically deployed, aluminum, telescoping tubes with titanium skid plates and is used to support the vehicle during landing. All four landing gears were extensively damaged during the fire and explosions. Portions of landing gears 3 and 4 were melted. Some internal pieces may be recoverable. (An extensive series of photographs of the landing gear may be found in Appendix G.)

SECTION 7

DATA ANALYSIS

7.1 BRIEF SUMMARY OF CLIPPER-GRAHAM, DC-XA, PROGRAM

McDonnell Douglas' Clipper-Graham, DC-XA, reusable vehicle was a technology demonstration rocket craft. Unlike conventional launch vehicles, the DC-XA was an autonomously controlled, single-stage, reusable design (no jettisonable stages) capable of airplane-like turnaround between flights. This vehicle was designed to vertically takeoff and land, using the thrust from its four throttlable engines to slow its descent, and to return for landing fully intact. The vehicle stood 43 feet high and was approximately 13.5 feet across the base.

Program objectives were to integrate advanced launch technology components into the DC-XA; to demonstrate performance, operability, and supportability of reusable launch vehicle (RLV) components through ground and flight testing; and to demonstrate rapid prototyping in a combined government and industry cooperative effort.

The DC-XA was built under a cooperative agreement between NASA Marshall Space Flight Center and McDonnell Douglas Aerospace (refs. 2 and 3). Key advanced technology components were integrated into the DC-XA at the aerospace firm's Huntington Beach facilities. McDonnell Douglas integrated a cryogenic composite liquid hydrogen tank, composite intertank structure, and elements of the gaseous oxygen and gaseous hydrogen reaction control system into the DC-XA. Several pieces of foreign technology were also used in developing this vehicle. A cryogenic liquid oxygen tank made from an advanced Russian aluminum-lithium alloy has was integrated into the vehicle. A German-built aluminum and Inconel landing gear with titanium skid plates was used to support the vehicle during its landings.

The DC-XA was a follow-on to the DoD's DC-X Single-Stage Rocket Technology program which completed a series of eight flight tests ending in 1995. The DC-X and DC-XA programs have demonstrated streamlined program management, rapid development of prototypes, and operation and maintenance of reusable launch vehicles. The DC-X was completed 18 months after the DoD awarded the contract to McDonnell Douglas and was transferred to NASA in 1995. The mishap occurred at the completion of the fourth flight. The objectives of the flight were to obtain additional systems knowledge, land on a concrete surface, and to investigate maneuvering dynamics.

7.2 SUBSYSTEM DATA REVIEW

7.2.1 GUIDANCE NAVIGATION AND CONTROL

Approximately 154 seconds of plotted guidance navigation and flight control data were examined from flight 4 of the DC-XA. Engine ignition occurred at 19:14:58.750 UTC (13:14:58.750 MDT), and ascent commenced 3.5 seconds later. The vehicle ascended to an altitude of 4050 feet as measured by the inertial navigation system (INS) (4200 feet measured by the radar altimeter) and achieved a maximum vertical velocity of 185 feet/second during the ascent. The ascent portion of the flight lasted approximately 40 seconds. Upon achieving altitude, a pitch maneuver was initiated that rotated the vehicle from a +17 degree pitch attitude to -41 degrees, back to +21 degrees, then returning to a 0 degree pitch orientation of the craft. This maneuver was accomplished in approximately 40 seconds, and the vehicle remained near this attitude throughout the rest of the flight. The pitch rate of the vehicle was maintained below 6 degree/second throughout this maneuver. The yaw attitude of the vehicle remained within a band from +2 degrees to -3 degrees, and the pitch rate was maintained below 2.5 degrees/second throughout the flight. The roll attitude of the vehicle remained between 163 degrees and 166.5 degrees throughout the flight, and no vehicle rates above +1.5 degrees/second and -2 degrees/second were achieved in this axis. The east-west velocity of the vehicle never exceeded 140 feet/second, and the north-south velocity never exceeded 31 feet/second during the maneuvers.

Descent was initiated at approximately 103 seconds into the flight and terminated at weight- on-gear at 19:17:24.560 UTC (13:17:24.560 MDT), an approximate descent of 39 seconds. The vertical velocity never exceeded 180 feet/second during this period.

During the ascent portion of the mission, a developmental test objective featuring actuation of four of the five flaps on the vehicle was performed. This test was done to measure the affects of the flaps on vehicle attitude and was not part of the control function. At 12 seconds into the flight, flap 3 was opened to 22 degrees, and flap 4 was opened simultaneously to 18 degrees. Both flaps remained open for 8 seconds then were commanded closed. At 20 seconds into flight, flap 2 was then commanded open to 15 degrees, remained open for 3 seconds, and then commanded closed. Flap 1 was commanded open at 28 seconds into the flight, remained open for 8 seconds, and commanded closed. All flap activity had ceased approximately 150 feet before achieving the 4050 feet maximum altitude. Only slight perturbations to vehicle attitude were seen when the flaps moved. Hydrogen flames from a vent behind flap 5 were visible during descent and were not deemed abnormal by the test team.

A loss of radar altimeter altitude data occurred twice during the flight. Both data dropouts happened when the vehicle pitch attitude exceeded +15 degrees. These dropouts were explained by the skewed orientation of the vehicle and beam width limitations of the instrument resulting in the beam not seeing the ground at the higher pitch attitude. The navigation system was able to recognize the loss of data and did not use it in the guidance control.

A region of increased flight control activity was observed on the engine actuator commands and position feedbacks approximately 100 seconds after launch. This activity occurred during descent at an altitude of 3500 feet. This engine activity correlated to throttle up of the engines to reduce this rate of descent. The vehicle control system maintained the vehicle orientation very well from this point through landing. Flight control rate gyroscope and accelerometer assemblies responded as expected during flight, measuring vehicle changes of attitude and showing no large or abrupt vehicle perturbations.

The vehicle electrical system currents and voltages were examined and showed no dropouts or unexpected excursions during flight. Data from a set of accelerometers mounted on the avionics panel were examined and did not show any unusual vehicle perturbations. Several data glitches were observed on the telemetry and data acquisition system, but did not appear to have any ramifications pertaining to the mishap.

In summary, of the data examined relating to flight control performance, electrical power distribution, and accelerometers (on the avionics pallet), no anomalies were observed that may have contributed to the mishap on flight 4 of the DC-XA.

7.2.2 INSTRUMENTATION

Instrumentation data were nominal until the time of landing gear deployment. Helium tank pressures and temperatures were similar to previous flights, except for the helium decay rates post-landing gear deploy. There are five temperature sensors in the helium system. Two sensors located in the intertank area ranged between -30˚ F and - 40˚ F. One was located in the avionics area (nose cone), and two were in the boattail. These three temperatures ranged between 20˚ F and 45˚ F. All five sensors were nominal. The pressurization to flight pressure of approximately 3400 psia was nominal. No dropouts or erratic behavior was seen with these pressure transducers in either the pressurization bottle helium system or the control bottle helium system.

There were two other pressure transducers in the control bottle system, downstream of a 750 psia regulator and a 500 psia regulator. Other than for a few pounds of pressure upward drift attributable to regulation creepage, these data were nominal. There was an additional temperature sensor installed on one of the helium shutoff valves that supplies pressure for landing gear deployment. This temperature sensor indicated a range of approximately 35˚ to 50˚ F. This range was in line with the temperature sensors located in the boattail.

Propulsion valve talkbacks were nominal. Hydraulic temperatures appeared nominal. External calorimeter data at the hydrogen vents and at the engine chilldown duct vents showed some activity as expected. Additionally, a calorimeter located on the vehicle baseplate showed activity, but this activity correlated with liftoff and landing as expected. All four engine oxygen control valves (OCV) and the fuel control valve electromechanical actuator temperature sensors read around 0˚F and were nominal, except for the engine 1 fuel control valve actuator which read 1100˚ F. However, this exception was an erroneous indication that had always existed and was attributable to the instrumentation. The oxygen and hydrogen pump housing temperatures were nominal, giving no indication of a problem. The engine 4 OCV delta pressure shifted low moments after engine start and appeared to be an instrumentation or signal processing problem and did not appear to affect flight. All other temperature, pressure, and pump speeds on the engines appeared nominal.

Landing gear indicators on landing gears 1, 3, and 4 acted as expected, though gear 3 was slow to deploy. Gear 3 took about 0.6 second longer to reach full extension than normal. The time from initial motion to full extension was approximately within normal limits.

In summary, the instrumentation system responses were all normal with the exception of no indication on landing gear 2 extention following the extend command. Indications were that landing gear 2 remained stowed.

7.2.3. PROPULSION

Cryogenic loading proceeded nominally starting with setting up the GSE configuration for fueling and pressurization of the helium bottles to flight pressure (approximately 3400 psia).

Fuel loading was initiated first per standard sequencing. Oxidizer loading was initiated when fuel reached approximately 15 percent. Loading to flight levels took about 25 minutes (nominal), with liquid oxygen and liquid hydrogen tanks reaching flight levels at about the same time, then replenish mode was initiated. During loading, the tank and feedline pressures, the vent valve and the fill and drain valve talkbacks were nominal. Monitoring of the purge inside the vehicle indicated there was less than 1 percent hydrogen concentration. Corrective action is taken if levels reach 4 percent. Chilldown of the engines began as expected at T-minus 65 seconds and T-minus 40 seconds respectively, for both liquid oxygen and liquid hydrogen. Chilldown terminated for 10 seconds on each engine to allow for dispersal of hydrogen and oxygen vapors before engine ignition. The initial engine ignition parameter checked out nominally, as well as the 3.5 second go/no go check of the start and abort limits, which were approximately in the middle of the limit range. During flight, the commanded positions of the fuel control valves and oxygen control valves matched very closely with the actual positions. Additionally, the engine power levels tracked as requested except for a known condition on engines 1 and 2. At low power levels, a shift in the chamber pressure occurs which affects the fuel control valve and turbine feedback control loop. This condition with engines 1 and 2 has been accepted and does not affect flight. Finally, at engine shutdown, the liquid oxygen pump spooldown rates were nominal compared to previous engine runs indicating acceptable shaft and bearing wear.

In summary, from the data reviewed on the propulsion system, no anomalies were observed that would have contributed to the incident.

7.2.4 PNEUMATIC SYSTEM

The helium system consisted of the pressurization system and the control system (see schematic in Appendix G). The two systems are used as their names would indicate. The pressurization system included twelve 2.66 ft^3 helium bottles and associated tubing, valves, fittings, regulators, and orifices. This system provided pressure to the liquid hydrogen and liquid oxygen cryogenic tanks and pressure for engine and fill/drain line purges and was a redundant source of pressure for deploying the landing gear. The control system consisted of one 2.66 ft^3 helium tank and associated hardware. It was used primarily to provide actuation pressure to valves, such as tank vent valves, propellant fill/drain valves, and engine valves. It also provided helium to purge the four engine interpropellant seal passages. Additionally, it supported a maneuvering or reaction control system (RCS) which was not operational for this flight but was flown inerted. Finally, the control system provided a redundant supply for extending the landing gear.

The helium control system provided regulated pressure to three subsystems. The 750 psia system provided the pressure to the tank vent valves, engine interpropellant purges, and auxiliary propulsion system. The 450 psia system provided pressure to the propellant fill and drain valves and to the engine control valves. Pressures in the 450 psia and 750 psia systems were normal during flight. The 160 psia system provided pressure to the landing gear and did not have a pressure measurement.

As discussed in the instrumentation section, system temperatures were nominal throughout flight. The helium pressures and decay rates also appeared nominal up until the point of landing gear deployment. When the landing gear deployed, the decay rates of these supply and control helium bottles increased. When compared to the previous flights, the decay rates were significantly higher, increasing by approximately 50 percent in the helium control system and 75 percent in the pressurization system.

In summary, all of the system data looked normal up until landing gear deployment. At that time, both helium bottle supply systems showed an unusual decay in pressure indicating an increased demand on the system. (See section 7.3.5.)

7.2.5. HAZARDOUS GAS

The aeroshell and boattail were purged and inerted with gaseous nitrogen starting at preflight. There were two hazardous gas detection systems. One system monitored the aeroshell interior for hydrogen indications. This system used a vacuum pump to pull samples of gas through a hydrogen detector on the ground. All of the indications were less than 1 percent concentration. The second system was an onboard experimental system provided by Marshall Space Flight Center that is still in the early stages of development. There were 32 sensors located throughout the vehicle monitoring hydrogen leakage during loading and flight. It was reported that one sensor in the boattail indicated approximately 35 percent concentration of hydrogen during loading. This report was discounted as an instrumentation problem for two reasons. First, other sensors nearby did not indicate any leakage. Secondly, the Ground Systems Manager monitored the purge for hydrogen, and the indicators were all less than 1 percent concentration. There were no indications in the hazardous gas system that would have contributed to the incident.

7.3 LANDING GEAR SYSTEM

7.3.1 FLIGHT OPERATION

Each landing gear consists of three telescoping cylinders located inside the protective aeroshell. (Schematics and drawings of the landing gear and pneumatic lines may be found in Appendix G.) The gear was deployed and retracted by a pneumatic actuator, powered by the vehicle's 160 psia helium supply system. It was held in place by a brake mechanism which was released by the pneumatic pressure. Once the landing gear was extended, it was locked in place by a spring-loaded downlock. The bottom cylinder was filled with crushable honeycomb and had a hydraulic damper within it to absorb any loads experienced during landing. There was a stow position sensor at the top of the gear and a potentiometer that indicated the extension of the gear. During landing when one of the landing gears indicated weignt-on-gear, engine cutoff was initiated by the Vehicle Management System Computer (VMSC).

Pneumatic pressure was supplied to all four landing gears by a common manifold. The helium supply system consisted of two systems. The pressurization system consisted of twelve 2.66 ft^3 bottles, and the control system which had one 2.66 ft^3 bottle. Each system fed a regulator which reduced the pressure from supply pressure (normally around 2600 psia) to 160 psia at the solenoid operated shutoff valve. The lines downstream of the shutoff valves were joined together to feed the common manifold. Pressure was supplied to the landing gear by energizing either of the solenoids. Typically, both solenoids were energized. The pneumatic line which ran from the manifold to the gear was split at the gear, providing pressure to the brake port and to the deploy or extend port on the gear.

7.3.2 FLIGHT DATA

During flight, the VMSC initiates the gear deploy command nominally 10 seconds before touchdown. The command opens both helium shutoff valves (SOV-4 and SOV-9), providing actuation pressure to release the brake at the landing gear at the brake actuation port and pressure at the deploy pressure port to force the landing gear down. Weight of the gear assists the extension. As the gear moves, it loses the stow position indicator, and the pickup of the full extension indication typically comes less than 1 second later.

The landing gear deploy command was initiated at 19:17:11.180 UTC (13:17:11.180 MDT). Landing gears 1 and 4 stow indications dropped out at 19:17:11.940 UTC (13:17:11.940 MDT) and 19:17:12.130 UTC (13:17:12.130 MDT), respectively. Landing gear 3 was slow to deploy, losing its stow indicator at 19:17:12.240 UTC (13:17:12.240 MDT) and picking up the down and locked indication at 19:17:12.280 UTC (13:17:12.280 MDT). Landing gear 2 did not deploy, and the stow indication continued to indicate stowed. The actual extend time for landing gear 1 was 0.7 second, landing gear 4 was 0.8 second, and landing gear 3 was 0.6 second; all were within the range of normal timing. The weignt-on-gear was picked up at 19:17:24.560 UTC (13:17:24.560 MDT), resulting in the engine shutdown at 19:17:24.680 UTC (13:17:24.680 MDT).

7.3.3. INSPECTION

The vehicle wreckage was inspected by the Board on August 5, 1996. Particular attention was paid to landing gear 2 pneumatic connections. The brake release pressure line was found to be not connected to the brake release port. The connection where the brake line connects to a tee in the deploy line also appeared to be loose. (See photographs in Appendix G.)

The landing gear 2 pneumatic system was removed from the vehicle wreckage on August 12, 1996. It was disconnected at the manifold and the interface plate within the gear. Upon close visual inspection, both the male and female parts of the fitting were covered with soot. Also, the threads on both sides of the fitting appeared to be in excellent condition. (See photographs in Appendix G.)

7.3.4 FITTING TESTS

The Marshall Space Flight Center, Materials and Processes Laboratory conducted a test with a 1/4 inch fitting similar to the landing gear brake fitting. When tension was applied at a connected fitting with all threads engaged, the body of the fitting failed, instead of the threads. With two threads engaged, the threads failed at a tension of 5100 pound-force. Post test inspection of the threads indicated that the threads had stripped. It was also the opinion of the laboratory that given the geometry of the brake release pneumatic line, the fitting would not be apt to vibrate loose in flight.

7.3.5. HELIUM DECAY EQUIVALENT ORIFICE ANALYSIS

A detailed review of the pneumatic helium control system for the DC-XA vehicle was conducted. During the first three DC-XA flights, the control bottle helium supply system pressure decay rates showed a steady decay rate up to the time of landing gear actuation. At this time, an approximately 30 psia drop and partial recovery of the pressure occurred as the landing gear pressurization lines were initially pressurized. After 1 second, the decay rate decreased to what it had been before the landing gear deploy command. On the fourth flight, however, there was an approximately 50 percent increase in the decay rate over that before the landing gear deploy command.

The pressurization system experienced a similar pressure profile. During the first three DC-XA flights, the decay rates before the landing gear deploy command were approximately equal to the decay rates after the deploy command. After the landing gear deploy command was issued on flight 4, the pressurization bottle helium system decay rate increased by approximately 75 percent over what had been seen prior to the deploy command.

At the Kennedy Space Center, a simplified analysis was performed to calculate an equivalent orifice for the increased decay. Two cases were run. The first averaged the decay rates after the landing gear was actuated on the first three flights, then the result was subtracted from the flight 4 decay rate. The equivalent orifice from this analysis was calculated to have a diameter of 0.174 inch. The second case excluded the pressurization bottle decay rate from flight 2 and used the average from flights 1 and 3. Flight 2 bottle decay rates were thrown out because the ullage was much greater in the propellant tanks (due to less loaded liquid oxygen and liquid hydrogen). The resulting increase in flow rates into the tanks skewed the decay rates. The equivalent orifice for the second case was calculated to be 0.215 inch in diameter. These calculated equivalent orifices compare very well with the actual diameter of the brake release line which was 0.187 inch. (The complete data analysis may be found in Appendix F.)

7.3.6. FUNCTIONAL TEST OF LANDING GEAR 2

Landing gear 2 was removed from the wreckage on August 12, 1996, and shipped to the McDonnell Douglas Aerospace in Huntington Beach, California. (See photographs in Appendix G.) On August 22, 1996, a functional test and partial disassembly of the gear was conducted. Board member Charles E. Harris witnessed the test. (A complete report of the test is given in Appendix E.) The objective of the functional test was to evaluate the three mechanical failure modes that could have prevented deployment of the gear if the pneumatic pressure lines were properly connected. Referring to the landing gear fault tree in Appendix E, the three mechanical failure modes are (a) mechanical failure of brake, (b) mechanical jam of LDG sliding surfaces, and (c) mechanical failure jam of pneumatic actuator. An initial examination of the gear verified that the brake was engaged. Using pneumatic pressure supplied by a pressure bottle, the brake was activated and released. After unsuccessfully attempting to deploy the telescoping landing gear with pneumatic pressure connected to the actuator, the end of the gear was manually pulled out approximately 24 inches to the point where the spring loaded downlock latches of the locking mechanism were encountered. These two tests eliminated failure modes a and b. The end cap of the actuator was removed, and the piston inside the actuator cylinder was examined. Pressure was connected to the actuator in the stowing configuration to investigate the condition of the piston. An obvious pressure leak around the piston head was found to be the malfunction that prevented the actuator from deploying the landing gear in the functional tests. The telescoping end of the gear was then manually pushed back into the stowed position, and the pneumatic line to the brake was vented, thereby engaging the brake. These actions verify that the actuator was not jammed, thereby eliminating failure mode c. A prior functional test conducted by McDonnell-Douglas showed that the landing gear would deploy under gravity loading only. (See documentation in Appendix E.) Therefore, if the brake line had been properly connected on the vehicle and released during the landing maneuver, the gear would have deployed even if the actuator was leaking. However, it is the view of the Board that the piston seal was damaged by the fire after the accident. This conclusion was reached because the actuator successfully deployed and stowed the landing gear during the load on gear test conducted during the preflight vehicle preparation.

7.3.7 SUMMARY

Considering the data analysis that indicated no abnormal vehicle performance, a higher decay rate in the pressure data following gear deployment command and the condition of the hardware, the Board concluded that the landing gear 2 brake release control line was not connected during flight 4. If the brake is not released, the landing gear will not extend even with full pressure applied to the extend port. Also, it was verified at McDonnell Douglas Aerospace in Huntington Beach after the mishap that landing gear 2 mechanical brake, landing gear sliding surfaces, and pneumatic actuator were not jammed and would have functioned as designed if properly connected.

7.4 PREFLIGHT LANDING GEAR OPERATIONS

7.4.1 PREFLIGHT OPERATIONS

Before every flight, a landing gear test is performed as part of the “DC-XA Systems Servicing and Preparation” procedure number DCXP-002 (ref. 1). The system is tested as part of the propulsion preparations which are normally run 1 to 2 days before the flight. The test is started by the Flight Operations Control Center (FOCC) which deploys the gear by activating two solenoid shutoff valves (SOV-4 and SOV-9) and verifies that the deployed indication is displayed in the FOCC. A mechanical technician manually raises each gear individually until the weignt-on-gear indication is seen in the FOCC. The FOCC then removes the deploy command by deactivating the solenoid shutoff valves, which activates the brake.

To retract the gear, three technicians are required: mechanical technician (MT) 1 at the aeroshell access door, MT2 inside the aft fuselage on the base heat shield, and MT3 located on the ground beneath the base heat shield. MT1 gains access to the aeroshell access door by using either a Marklift work platform or a ladder. MT1 removes the access door and reaches in to disconnect the brake pneumatic connections. Then MT1 connects a flex hose from a ground support service panel to the brake line and the gear up connection. The gear down line is left open to vent.

Then, MT3 removes two safety locking screws. This removal allows the unlocking cylinder to rotate. MT1 opens a hand valve to apply pressure to the brake port, thereby releasing the brake. MT2 rotates the unlocking cylinder to retract the spring-loaded downlock latches. MT1 then opens the valve on the service panel to pressurize the upline to raise the gear to within 18 inches of the base heat shield. Then MT1 closes the valve to the brake line and vents the pressure to lock the gear in place. MT2 rotates the locking cylinder to re-arm the downlock latches. The cylinder is then rotated to a position where the safety locking screws are lined up. MT3 then installs the screws hand tight to prevent the unlocking cylinder from rotating in flight. MT1 pressurizes the brake to release it and the gear continues to raise to the stowed position. As the gear is being raised, MT3 ensures that the alignment pins on the feet line up with the holes in the base heat shield. MT1 then closes the brake hand valve, vents the brake line to ambient, closes the hand valve on the upline, and vents this line to ambient. A verification is received from the FOCC that the stowed indication appears for the affected gear. MT1 then disconnects the ground supply pneumatics from the brake port and the upline port and reconnects the brake flight pressurizing assembly. The access door is reinstalled. This process is repeated on the other three landing gears.

Once the gear has been stowed, there is no method to verify that it is in the proper pneumatic system configuration. If the line is pressurized on the ground, the brake would be released and the landing gear would deploy under its own weight. The design does not provide a method of testing the pneumatic connections.

7.4.2 FLIGHT 4 PREFLIGHT OPERATIONS

During preflight preparations for DC-XA flight 4, the landing deployment and weight-on-gear indication test was successfully performed on all four gears the afternoon of July 29, 1996. Because it was late in the day, the landing gear stow operations were delayed to the following morning.

On the morning of July 30, 1996, mechanical technicians came to the vehicle to stow the landing gear. The Marklift work platform was being used by an avionics technician and an avionics engineer to service the avionics rack near the top of the vehicle. Gear stow operations did not get underway until 14:00 MDT, and this was the only task left to be performed in the DCXP-002 procedure. The mechanical technicians started on gear 3, and the stow operation went smoothly. They then moved to gear 2. The stowing operation started well. After the technician disconnected the ground pneumatic lines from the brake and the up ports, an avionics technician came back with a request for the Marklift work platform to complete his task at the avionics rack at the top of the vehicle. Also, the Flight Engineer warned the technicians to stay clear of the flaps since hydraulics might be activated to center the engines to support the Optical Plume Anomaly Detection System (OPADS) calibration by NASA Marshall Space Flight Center engineers.

At this point, the statements obtained during the interviews vary. One technician stated that the landing gear reconfiguration task was completed and the access door installed before the Marklift work platform was relinquished to the avionics technician. Another statement recalled the warnings about hydraulics but stated that the technician came down and gave the Marklift work platform to the avionics technician. When the avionics technician completed his task 15 minutes later, he states the mechanical technician went back up on the Marklift work platform to install the access door.

Another mechanical technician, who was in the boattail supporting the gears 2 and 3 retraction by rotating the unlocking cylinder and was working on the hydraulics system, stated hydraulics could not have been brought up because he had reduced the accumulator and reservoir pressures to disconnect the auxiliary power system hydraulic lines.

The Marklift work platform was then moved around to the other side of the vehicle where access could be gained to landing gears 1 and 4. Landing gears 1 and 4 were stowed with no interruption.

As a final check, the Flight Engineer then requested status of the landing gears from the FOCC and was told that all the stow indications were on. The Propulsion Engineer asked the technician if all lines were connected and was told yes by the technician.

The Board could not recreate the actual sequence of events for hydraulic operations, recorder activation, and landing gear stowage indication because there was no recording of the communication channels, there were no log books, and the data tapes either could not be located or had been written over. Based on the interviews, there was a significant amount of activity going on in the area when landing gear 2 was being stowed and closed out for flight.

7.4.3 NORMAL SEQUENCE

Normally the sequence employed in running the DCXP-002 procedure is to do propulsion checks followed by landing gear checks then hydraulics check. The hydraulic technician reported that after he serviced the system and brought up hydraulic pressure, it was his practice to crawl into the forward compartment above the bulkhead and inspect the hydraulic lines. While in this area, he performed an informal inspection of the area including the landing gear pneumatic lines.

Before DC-XA flight 4, the hydraulics system was checked out of the usual sequence before the landing gear was checked and stowed. The hydraulics technician had already checked for leaks and did not see a need to repeat it. Because the hydraulic check was done out of the usual sequence, the discretionary survey of the area which might have revealed the disconnected brake line was not performed. (The “as run” procedure, see Appendix D, showed no indication of procedural step completion for gear stowage.)

7.4.4 DC-X FLIGHT 3 INCIDENT

During the interview process, it was discovered that this failure to connect the landing gear brake line was not an isolated incident. While preparing for DC-X flight 3 in 1993 prior to turnover to NASA, a preflight inspection was being performed by a hydraulics technician above the bulkhead. He discovered that the landing gear 2 lines had not been connected. It was called to the attention of the other technicians and corrected before flight.

Although the problem did occur there was apparently no documentation taken on the incident. The technicians reported they had a higher sensitivity to this connection, but no changes were made to the procedure to highlight the issue. This appears to have been a missed opportunity to tighten up the controls of this critical process.

7.4.5 SUMMARY

The Board concluded that the design of the landing gear pneumatic control system was a poor design. It required that a system that had been tested in its flight configuration be disconnected and reconfigured in order to stow the landing gear. The pneumatic system lines were then returned to the flight configuration with no pneumatic test or inspection to verify that the gear was properly configured in the stowed position.

The Board also concluded that the mechanical technician, while reconfiguring the landing gear pneumatics, may have been distracted or interrupted by the events at the time he was about to connect the brake pressure line. In any case the connection was not accomplished.

7.5 SYSTEM SAFETY ANALYSIS

The Board requested that McDonnell Douglas Aerospace (MDA) provide all system safety analyses that had been conducted on the DC-X and DC-XA vehicles. Two documents were provided. The first was titled "SSRT System Operational Hazard Analysis Report," document number AAT-RML-SSRT-012X, and dated September 25, 1992 (ref. 4). The second document was titled "DC-X Vehicle (Experimental) Functional Hazard Analysis Report," document number AAT-RML-SSTO-005X, revised November 29, 1991 (ref. 5). The first document was not applicable to the DC-XA mishap. The functional hazard analyses outlined in the second document went only to the system level. Neither analysis was of sufficient depth to address the circumstances that led to the DC-XA mishap. In addition, the Board independently obtained some additional information from the WSMR government files. This information was very sparse and did not provide an indication of any system safety analyses to a depth sufficient to address the circumstances leading to the DC-XA mishap.

In addition, limited hazard analyses (ref. 8) and a System Failure Modes and Effects Analysis (FMEA) (ref. 9) were conducted on the DC-X vehicle. The FMEA does denote the failure condition "Landing Gear Does Not Extend" but does not address potential subsystem or component failure modes that could induce the condition. Consequently, it could not be used to isolate single point failures (SPF) within the landing gear system. A SPF should require some failure mitigation action, such as design change, redundancy, or inspections.

7.6 CONFIGURATION MANAGEMENT AND WORK CONTROL

7.6.1 OVERVIEW

The DC-XA was designed, built, and operated under the MDA Rapid Prototyping Process Guidelines. The concept was to develop a single stage rocket technology/single-stage-to-orbit (SSRT/SSTO) vehicle to achieve an aircraft-like turnaround demonstration and provide lessons learned for future operational vehicles.

The rapid prototyping process had as an objective to significantly reduce paperwork and personnel requirements and to significantly reduce design and operational procedures. Inspection requirements were limited to only those identified by Engineering or Quality Assurance and identified or specified on the design or development drawing.

7.6.2 CONFIGURATION CONTROL

The design of the DC-XA vehicle is controlled by drawings released by Engineering Orders (EO). The rapid prototyping process allows a flexibility for engineers to develop and document hardware in realtime. Drawings are less formal then production releases and may contain relines, sketches, or photos.

At the WSMR, the Chief Engineer served as the configuration manager. Any configuration changes were controlled by EO's signed by the Chief Engineer. The work to implement changes was released using the supportability information form (SIF) initialed by the Chief Engineer and the Crew Chief. Once completed, the configuration change SIFs are maintained by Quality Assurance.

7.6.3 PRELIMINARY REVIEW DOCUMENTS

Hardware found to be nonconforming is documented on a Preliminary Review Document (PRD). All diagnostics and failure analysis was performed using the PRD which contained the results on acceptability. The action could result in rework or possibly a use-as-is disposition. The PRD's are generally written by Quality Assurance, dispositioned by Engineering with Quality Assurance approval. There were very few of these documents and they appeared to be well written.

7.6.4 WORK CONTROL SYSTEM

All work performed on the vehicle, the ground support system, and the FOCC was authorized by a supportability information form. The SIF is analogous to a work order. It was developed as part of the rapid prototyping process to gather data on the work accomplished such as time to perform the task and the manhours required to accomplish the task. The SIF can authorize procedures or partial procedures, special tests and inspections, or modifications to the vehicle. The form is prepared by MDA Engineering or Quality Assurance and nominally approved by the Chief Engineer, the Quality Assurance manager, and the Crew Chief. The approved SIF is sent to the Crew Chief who schedules the work. The technicians perform the work and annotate the start time, elapsed time, number of people involved, any reasons for work stoppage, and the completion time. If an anomaly or discrepancy occurs, it is to be documented on the SIF and returned to Engineering for disposition. The Chief Engineer and Quality Assurance must approve the disposition before it is sent back to the Crew Chief for work.

The closed and open SIFs dated from 1/20/96 to the time of the mishap on 7/31/96 were provided to the Board. A review of the SIF's indicated that they were generally adequate to perform the task. However, there were at least several cases where tasks were known to have occurred, however the authorization could not be found in the SIF's provided. Examples of the tasks are: (a) the troubleshooting that was being performed on the camera recorder on the morning of 7/30/96, (b) the disconnection of the hydraulic lines from the APS, and (c) portions of the DCXP-002 procedure. The implication of this review is that at least a portion of the work control system was undisciplined and work was being performed on the vehicle without documentation.

7.6.5 OPERATIONS

All of the operations, checkouts, and flight of the DC-XA are defined in four procedures.

DCXP-001	Setup and Checkout of Facilities.
DCXP-002	Systems Servicing and Preparation.
DCXP-003	Preflight Procedures (ref. 6).
DCXP-005	Postflight Procedures (ref. 6).

A fifth procedure, DCXP-004 (ref. 6) titled, "Inflight Emergency Procedures", deals with problems that may come up during launch operations or the flight. Procedure numbers 002, 003, 004, and 005 are performed under the direction of the Flight Manager in the FOCC.

The Board was told that the DCXP-002 procedure was run in its entirety prior to flight. This could not be confirmed by reviewing the SIF's. Sections of the procedure could not be found in the closed documentation.

7.6.6 TEST PREPARATION SHEETS

Commodities used by the DC-XA were provided by NASA through the White Sands Test Facility (WSTF). The procedure DCXP-002 calls out the use of JSC Test Preparation

Sheets (TPS) to perform the replenishment of the fluids. WSTF personnel would perform these operations. These TPS's were not reviewed since they would not have contributed to the mishap.

7.6.7 <u>TEST CARDS</u>

Flight tests are conducted by means of test cards. A review of the flight test no. 4 card indicated good control of the operation. (See Appendix C.) The test cards define the test objectives, identifies the flight crew, and details any changes to the DCXP-003 and DCXP-005 procedures. It defines and requests verification of the current software version for both the flight and ground systems. The card contained a checklist which is used by the Flight Manager to assure a well-defined process proceeding and following a flight.

7.6.8 <u>SUMMARY</u>

The work authorization and control system for performing day-to-day tasks is somewhat lax, with the exception of the flight test. Tasks which should have been performed were not found in a review of the SIF. Work on the vehicle was being performed which was not documented. The rapid prototyping process significantly reduced inspection requirements. The operation of stowing the landing gear was not identified as a critical process, and therefore, an inspection was determined to be not required. The vehicle perparation process required an intimate knowledge of the subsystems by the technicians performing the work. While this may be somewhat consistent with the rapid prototyping concept, it could lead to missing or overlooking a task. The philosophy of demonstrating aircraft-like turnaround capability with minimal written procedures went beyond what is considered acceptable for a research vehicle.

7.7 PROCEDURES ANALYSIS

7.7.1 <u>PROCEDURES REVIEW</u>

A review of the four procedures (refs. 1 and 6) which are used to test and operate the DC-XA was performed. Particular attention was paid to the "DC-XA System Servicing and Preparation" procedure number DCXP-002. While the procedure generally is adequate to perform the checkout and launch of the vehicle, a number of deficiencies exist. These procedures include a pretask briefing, task objectives, sufficient support information and the required steps to perform the operation. However, the deficiencies lie in the areas of procedure clarity, organization, verification steps, and configuration control management.

For example, the procedure for stowing the landing gear was not specific. Several steps contained multiple tasks within the step. Without discrete verification steps, tasks could be overlooked. Also in reviewing the method of stowing the gear with the technicians and engineers, it became apparent that the actual method of stowing the gear was not the same as the steps in the procedure. Although the modified ad hoc method used appeared acceptable, it should have been documented in the existing procedure.

There was not a way to verify after the fact that the steps in a procedure had been performed. As a procedure is being performed, the Flight Engineer or Propulsion Engineer on the ground may have the procedure in hand, but the technician performing the work in the vehicle or on the Marklift work platform does not. Typically in the aerospace culture, as a step is performed in a procedure, it is either checked, signed, or stamped by a technician or inspector as having been performed. This notation was not done on this program. Such notation would have provided a control assuring that all the steps have been completed as well as a historical record of the work performed.

There is a sequence in DCXP-002 entitled Task 3-07F-1 which opens the landing gear system access doors and visually inspects the electrical harnesses and pneumatic plumbing. It could not be verified by looking at the SIF whether this sequence was ever run. It is also unclear when this sequence would be run (i.e. before the landing gear stow sequence or after). It could have been intended to open the panels and perform the inspection prior to the landing gear functional test and subsequent stowing operation.

In reviewing the DCXP-002 and the SIF, a step to install the access panels could not be found. A SIF was found which was written by Quality Assuranceto perform an inspection on the day of flight which included a visual verification that “all doors/panels are closed/secure”.

The method used to close out a procedure was to close the SSRT Supportability Information Form (SIF). The SIF was closed on the landing gear stow procedure for DC-XA flight 4 on July 29, 1996, but the task was not completed until July 30, 1996. In reviewing the SIF, it was not apparent that the entire procedure DCXP-002 was performed.

Because of its large volume of tasks and data, procedure DCXP-002 creates confusion and discontinuity. The procedure could have been divided into several smaller more concise documents. The procedure also includes a number of references and additional procedures which complicates the task.

7.7.2 SUMMARY

The procedures were adequate in basic content and direction but need improvement. The work control system indicated a lack of discipline in preparation and close out of the procedures and Supportability Information Forms. The technicians had a high level of knowledge of the DC-XA and based on their experience, at least in one case, developed an alternate procedure which was not reflected as a procedure update. This alternate, undocumented procedure could have also led to confusion.

7.8 WEATHER CONDITIONS

The weather during the day of the operation was warm but not unusual. (All weather data is given in Appendix H.) On July 30, 1996, there were light and variable winds with gusts to 11 knots, and it was clear with low humidity and temperatures in the upper nineties. Weather was determined not to be a factor during the gear stowage operation.

At the time of the mishap, weather conditions were perfect for the launch. The weather was sunny, clear, and there was an 11 knot breeze from south-southeast. The program required the wind to be below 12 knots. Weather was determined not to be a cause of the mishap.

7.9 HUMAN FACTORS

7.9.1 LANDING GEAR INSTALLATION TEAM

The landing gear stow team consisted of a crew of five, the Propulsion Engineer and four technicians, qualified, trained, and current in all aspects of their job. All technicians were trained and qualified in the DCXP 002 procedures and able to perform all functions required of this task. Mechanical technician (MT)1 was on Marklift work platform approximately 15 feet off the ground. MT2 entered the boattail through the access panel behind the flap for access to the landing gear's unlocking cylinder. MT3 and MT4 were below the launch vehicle helping to position the foot pad onto the pins of the bottom of the heat shield and to remove and install the landing gear's unlocking cylinder set screws. The engineer watched the entire procedure, questioned the technicians on the completion of the tasks, and verified proper gear stowed electronic sensor readings over the net with the Ground Systems Manager in the FOCC.

The technicians and engineer had been with the program since the program's transfer to NASA from the Air Force. All had started in their current jobs while the program was under Air Force program management as the Delta Clipper (DC-X). All were very familiar with the intricacies of their job, the configuration of the hardware, and operations procedures.

During the week of the mishap, the pad crew along with the rest of the team was configuring the vehicle for launch. The team worked 6 days of 10 hours per day for the 2 weeks before the launch. Of this 10 hour work day, approximately 1 hour was spent driving from the Small Missile Range Gate to the site and back.

On the day of the maintenance problem, all McDonnell Douglas personnel were locked out from the site for range operations until approximately 09:00 MDT. Range lockout was a common occurrence. Thus, the gear stowage procedure began at approximately 14:00 MDT in the afternoon and was completed by 16:00 MDT for all four landing gear. The work day and conditions were not out of the ordinary.

7.9.2 SUMMARY

The human factor that could have influenced the mishap is assumed to be distraction of the mechanical technician by other maintenance operations being performed by fellow McDonnell Douglas technicians and engineers as well as NASA Marshall Space Flight Center personnel. During the 2 hours the technician was performing the gear stowage procedure the following other major tasks were happening or about to happen: OPADS, Auxiliary Propulsion System (APS) disconnection, recharging the high-pressure helium and nitrogen ground supply tanks, and avionics camera troubleshooting. Confusion and conflicts arose over the use of the Marklift work platform and personnel.

The hours that the team worked were not excessive. The procedure began at 14:00 MDT and lasted until 16:00 MDT on a weekday. The technician working the landing gear pneumatic connections averaged approximately six 10-hour days per week before launch. He reported that he felt normal and was not stressed at the time the procedure was run.

7.10 SUMMARY OF CONTROL ROOM ACTIONS

7.10.1 ABORT MODES

Three basic abort modes are available to the Flight Manager. The first is the auto climb. This mode commands a vertical velocity of 200 fps and nulls lateral velocity. The vehicle climbs to 7000 ft above ground level where the three parachutes can be deployed. At parachute deployment, the engines are shutdown, and the liquid oxygen tank is vented.

The second is auto translate which commands the vertical velocity to zero in a hover mode. The mode commands an altitude of 300 feet if the vehicle is below 350 ft. It chooses either the landing pad or a point 1650 ft uprange whichever is closer when the command is executed.

The third is auto landing. This mode nulls the lateral velocity, limits vertical acceleration to ±5 fpss, and deploys the landing gear. When lateral and vertical velocities and accelerations are within limits, the nominal landing phases are sequenced as in a normal landing. During this abort, there is no position steering, so the vehicle will not return to the landing pad but will land at a position near where it is when the automatic land command is given. This abort mode was successfully executed on DC-X flight 5 when a hydrogen detonation damaged a flap shortly after take-off.

7.10.2 CONTROL ROOM ACTIONS

The Flight Operations Control Center was manned at 09:00 MDT on August 31, 1996, to begin preflight operations procedure DCXP-003. The crew consisted of the Flight Manager, Deputy Flight Manager, Ground Systems Manager, and the Flight Engineer. The flight sequence application was started at 09:02:08 MDT. At 09:38:33 MDT, a simulated flight was performed. Propellant loading was started with liquid hydrogen at 12:27:45 MDT followed by liquid oxygen starting at 12:28:03 MDT.

At 13:07:26 MDT, the automatic flight sequence was activated. Once this sequence was initiated, the engines would ignite in 120 seconds. At this point, the vehicle was automatically controlled. Engine ignition occurred at 13:14:58.750 MDT, and ascent commenced 3.5 seconds later after the engines were start checked. During the flight, the Flight Manager watched a screen displaying the flight operation and was primarily interested in guidance but was also responsible for emergency procedures and abort actions if required. The Deputy Flight Manager monitored the vehicle subsystems.

Shortly after liftoff during engine throttle up, there was a flashing red and yellow light on hydraulics system 1. The Flight Manager called out the indication, and the Deputy Flight Manager had already checked and knew it was a known condition and responded that “we’re okay”. The remainder of the flight was nominal in all respects until the flight reached a point where the landing gear was deployed. The Deputy Flight Manager was watching for the indications of gear deploy on his display with the vehicle around 400-500 feet above ground level. He saw two gear deploy and after a hesitation saw a third deploy indication. After a brief delay and not seeing the fourth indication, he made the call “missing a gear.” At this point, the Flight Manager could elect to abort the landing or continue to land. He chose to allow the landing to proceed.

The Flight Manager’s rationale for his course of action was twofold. First was his reaction time. Data which were being displayed were about a second old, so the vehicle was closer to the ground than was being displayed. To execute the abort, he has to execute two commands, the arm and execute commands. These commands must be initiated before touchdown; otherwise, the engine will shutdown with the indications of weight on any gear. Once the auto climb command gets through, the vehicle modes out of landing into auto climb and the weignt-on-gear indication is not active. If the sink rate could not be reversed, the vehicle would impact the ground with the engines running. With such close timing, it could not be assured that the abort would be successful. There have been a number of simulations of attempting an auto climb at the landing gear deploy time. Some were successful and others were not resulting in vehicle impact. If the sink rate was not successfully overcome, the vehicle would hit the ground with engines running probably increasing the severity of damage. As a result, it was determined that the best resolution was to let the vehicle land at a slow velocity, around 4 ft/sec, shutdown the engines, and risk tipover.

The second reason is that if the auto climb were successful, the vehicle achieved the required 7000 ft above ground level and the parachutes were deployed, severity of the ground impact would depend upon dynamics of the vehicle swinging on the parachutes. The vehicle descends at a minimum of 30 ft/sec at an empty vehicle. The descent rate would be higher with propellant onboard. If the vehicle hit the ground at a high descent rate (30 ft/sec on parachutes versus 4 ft/sec with engines) and at an unknown angle, there probably would be more damage to the aft end of the vehicle.

7.10.3 SUMMARY

The Board concluded that the correct action was taken by not aborting during the landing phase of Flight 4 with a partial landing gear deployed situation. With the options available, the Flight Manager probably selected the least damaging one.

Having a planned emergency procedure to use in the event of “gear fails to extend” might have reduced the damage, but discussions with MDA personnel indicate that the failure was an accepted risk.

SECTION 8

CAUSES, FINDINGS, OBSERVATIONS AND RECOMMENDATIONS

8.1. PRIMARY CAUSE

8.1.1. Landing gear 2 brake release helium pneumatic line was not connected, preventing the brake from being released when the extend command was sent. This nonconnection led to vehicle instability upon landing and ensuing fire and explosions.

Findings:

a. Verified by inspection of pneumatic system in wreckage (photograph in Appendix G).

b. Substantiated by analysis of pneumatic pressure data (see paragraph 7.3.5).

8.2. CONTRIBUTING CAUSES

8.2.1. Design placed an unusual burden on ground operations by breaking connection and violating integrity of a just verified landing gear system.

Findings:

a. Design required manual disconnection of verified brake line to stow gear (see paragraph 7.4.1).

b. Integrity of stowed landing gear deployment system could not be verified with pneumatic instrumentation (see paragraph 7.4.1).

8.2.2. Landing gear stowage not identified as critical process.

Findings:

a. Limited hazard analyses, no critical item lists, and limited failure modes and effects analyses (FMEA) were inadequate to identify reconnection helium brake line as a critical process. (See paragraph 7.5.)

b. No critical sequences flagged in DC-XA Systems Servicing and Preparation (DCXP-002) Procedure (Appendix D, Vehicle Preparation Records).

c. No action taken to prevent reoccurrence after near miss on same procedure prior to vehicle turnover to NASA during DC-X flight 3 preparations. (See paragraph 7.4.4).

8.2.3. <u>No sufficiently detailed procedures check off process or cross-checking during the gear stowage line reconnection operations.</u>

Findings:

a. There was no separate step to close out the access panels. (See paragraph 7.7.1.) Detail in procedure was not sufficient to ensure consistent and complete ground operations. Multiple operations were included in a single step.

b. Undisciplined vehicle preparations by the ground team were dependent on technician's systems knowledge and checklist memory, informal cross-checks and availability of personnel and location of ground support equipment . (See paragraph 7.6.4.)

c. The "as run" procedure in Appendix D showed no indication of step completion for gear stowage. It is common aerospace practice to check off, initial, or stamp completed procedural steps. (See Paragraph 7.7.1.)

d. There was no second set of eyes required to verify configuration for closeout and access panel installation. (See Paragraph 7.6.1.)

8.2.4. <u>Maintenance technician was distracted and/or interrupted during gear stow operations.</u> This factor may have contributed to the nonconnection of the brake line by the maintenance technician.

Findings:

a. Conflicting testimonies on nature of the distraction or interruption. It is clear that the step was not completed (see paragraph 7.4.2).

b. Conflicting needs for the single Marklift work platform on the day of landing gear stowage may have caused an interruption. (See paragraph 7.4.2) The Marklift was shared for all upper vehicle maintenance processes.

8.3. OBSERVATIONS:

8.3.1. The McDonnell Douglas rapid prototyping guidelines or implementation of the guidelines may go too far in the direction of sacrificing quality and reliability.

8.3.2. Responsibility and accountability were shared by the team members at the launch pad. Although this sharing probably represented empowerment of employees, it appeared that no one was clearly in charge of pad operations.

8.3.3. The Flight Operations Control Center communicates with the pad by radio. Because these transmissions were not recorded, the Board was unable to accurately reconstruct the sequence of events around this mishap. In a future mishap this might prevent determination of cause.

8.3.4. All Flight Operations Control Center indications were recorded on data tapes during all prelaunch and launch actions. Prelaunch indications were overwritten within days of procedure accomplishment. Board was unable to reconstruct the sequence of events surrounding the gear stowage step completion.

8.3.5   The Range Hazardous Response agencies did not have the appropriate Hazardous Material Safety Data Sheets for the hazards associated with this accident prior to launch.

8.3.6.   The decision not to abort made by the Flight Manager was correct. There was no written emergency procedure for a partial gear deployment, and calling an abort would not have helped the situation.

8.4. RECOMMENDATIONS:

8.4.1. Critical procedural steps should be identified during systems design and flagged as critical in vehicle operations procedures. Then, independent verification of all critical steps should be performed during execution of operations procedures.

8.4.2. NASA should perform a handover design review when any program is transferred from another agency.

8.4.3. The "rapid prototyping" philosophy was cited as the rationale for employing minimal written procedures. The concept should be revisited from an operations perspective.

8.4.4. Prelaunch processing documentation and data tapes should be kept as historical records for each flight at least until a mission is completed and degree of mission success is understood.

8.4.5. Up-to-date hazardous materials information should be supplied to the appropriate hazardous response agencies at the start of any flight program in the future.

SECTION 9

DEFINITIONS OF TERMS AND ACRONYMS

APS -- auxiliary propulsion system

CIL -- critical items list

DCX -- Delta Clipper - Experimental

DC-XA -- Delta Clipper - Experimental Advanced (commonly called Clipper Graham)

F -- Fahrenheit

FCV -- fuel control valve

FMEA - failure mode and effects analysis

FOCC - Flight Operations Control Center

fpss - feet per second per second

ft -- foot

ft^3 - cubic foot

GSE -- ground support equipment

INS -- inertial navigation system

KSC -- Kennedy Space Center

MDT -- Mountain Daylight Savings Time

MSFC - Marshall Space Flight Center

MT -- mechanical technician

NASA - National Aeronautics and Space Administration

OCV -- oxygen control valves

OPADS -- optional plume detection system

psi -- pounds per square inch

psia -- pounds per square inch absolute

RCS -- reaction control system

sec -- second

SIF -- Supportability Information Form

SOV -- shut off valve

SSRT -- Single-Stage Rocket Technology

UTC -- Universal Time Coordinated

WSMR -- White Sands Missile Range

SECTION 10

REFERENCES

1. McDonnell Douglas Aerospace Company, DC-XA System Servicing and Preparation--DCXP-002, Revision B, dated March 25, 1996

2. Cooperative Agreement Between McDonnell Douglas and Marshall Space Flight Center, dated July 18, 1994

3. Modification to Cooperative Agreement Between McDonnell Douglas and Marshall Space Flight Center, dated February 4, 1996

4. Single Stage Rocket Technology System Operational Hazard Analysis Report, Document Number AAT-RML-SSRT-012X, dated September 25, 1992

5. DC-X Vehicle (Experimental) Functional Hazard Analysis Report, Document Number AAT-RML SSTO-005X, dated November 29, 1991

6. McDonnell Douglas Aerospace Company, DC-XA Flight Crew Operating Manual, Operating Procedures--DCXP-003, 004, 005, dated April 8, 1996

7. McDonnell Douglas Aerospace Company, Single Stage Rocket Technology (SSRT)/Single-Stage To Orbit (SSTO) Rapid Prototyping Process Guidelines, dated September 1994

8. Hazard Analysis for Single Stage Rocket Technology (SSRT), DC-X Program, Document Number, SSRT-SF01, dated July 15, 1993, Revision 1

9. Single Stage Rocket Technology System Failure Mode and Effect Analysis Report, Document Number AAT-RML-SSRT-008X, dated July 30,1992

VOLUME IV

LESSONS LEARNED SUMMARY

1. NASA should do a handover design review when any program is transferred from another agency.
2. System integrity should not be violated after a system is verified for flight, unless integrity can be reverified after reconnection.
3. Critical processes should be identified based on appropriate fault tree analysis, failure modes and effects analyses, and/or critical items lists. A second set of eyes should verify the results of critical steps and assure proper system configuration before closeout of an area.
4. A near miss should be documented and the cause of the near miss corrected.
5. Although procedures writing is an art, a procedural step should not contain too many tasks. In particular, tasks defined as "critical" should be broken out into separate steps.
6. There should be a disciplined way of physically checking off steps in a procedure.
7. A designated leader on the pad team should be responsible and accountable for task completion.
8. Pad technicians should be on a recorded communications net around a large vehicle for improved communications, to document the start and completion of procedural events, and flag anomalies.
9. In the design phase of a project, attention should be paid to vehicle access for maintenance. Specially adapted ground support equipment may be required.
10. Operations aspects of rapid prototyping deserve special consideration to make sure that safety and reliability of vehicle preparation operations are not violated.
11. Documentation, data tapes, and recorded voice tapes should be retained as historical records for each flight at least until a mission is completed and degree of mission success is understood.
12. The project should assure that up-to-date hazardous materials for a flight vehicle is supplied to the range hazardous response agencies at the start of any program.
13. NASA should consider means to maintain a suitable level of reliability and quality in cooperative efforts involving advanced concepts.